

# Large Deployable-Mesh Antenna System for Ocean Salinity and Soil Moisture Sensing<sup>1</sup>

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**Abstract**—A concept has been studied for remote sensing of sea surface salinity and soil moisture from space using a large deployable mesh antenna system. The antenna has a 6-m-diameter offset-fed parabolic reflector with multichannel feedhorns operating at L and S bands. The antenna boresight is offset from nadir and the entire system rotates about the nadir axis, providing a conical scan with high-precision measurements across a wide swath at a spatial resolution of about 40 km from a 600-km orbit altitude. The study includes evaluation of deployable mesh antennas and preferred antenna, spacecraft, and launch vehicle configurations. Designs for compact, lightweight feedhorns and radiometer/radar electronics are being developed and evaluated. Laboratory measurements of the emissivity of mesh samples under simulated orbital conditions are being made to evaluate the suitability of current mesh designs for high-precision radiometric applications.

large, lightweight mesh antennas to the remote sensing of land, ocean and cryospheric phenomena where low frequencies and/or high spatial resolution are required. Sea surface salinity and soil moisture are primary applications of this technology since they require low frequency observations at 1.4 GHz and spaceborne measurements of these parameters do not currently exist. This study has used these applications to focus the technology issues and to design and evaluate a baseline spaceborne system. For convenience, the instrument concept has been named OSIRIS (Ocean-salinity Soil-moisture Integrated Radiometer-radar Imaging System). However, the technology is potentially applicable to other remote sensing applications that can benefit from high-resolution microwave observations using frequencies as high as 37 GHz, such as ocean winds, precipitation, sea ice, and snow. Such observations are needed for detailed modeling and forecasting of the Earth's hydrologic cycle.

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## 1. INTRODUCTION

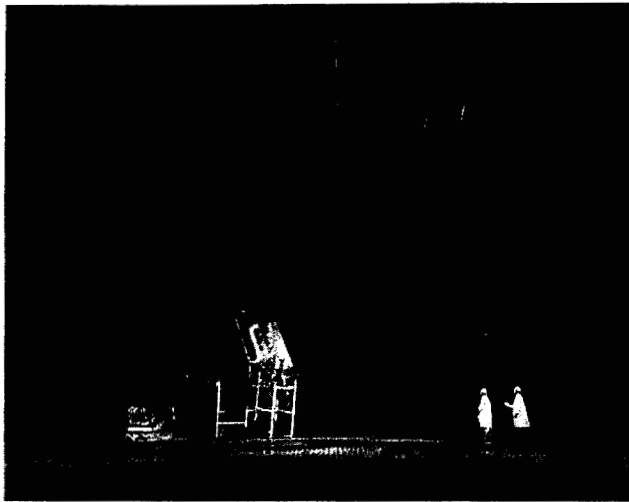
This paper describes a study of a large mesh antenna system for high resolution microwave real-aperture sensing of the Earth surface. The objective is to apply the technology of

The OSIRIS baseline system studied here is a 6-m-diameter reflector with two dual-polarized L/S-band feedhorns shared between 1.41 and 2.69 GHz radiometers and a 1.26 GHz radar. The 6-m antenna size was driven by a requirement for approximately 50 km spatial resolution or better from a 600-km orbit. However, many trade-offs are possible, and the benefits and impacts of antenna sizes in the 4- to 10-m range are also being considered.

The key technology is the lightweight mesh antenna. Deployable mesh antennas are a mature technology with extensive flight heritage in space telecommunications. Recently, 12-m-diameter antennas have been developed and

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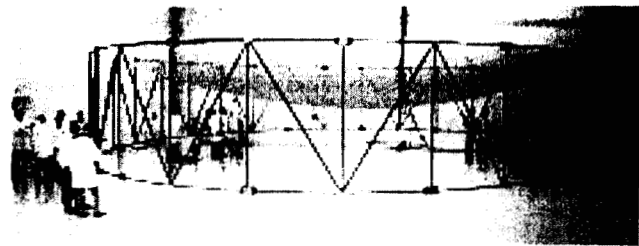
space qualified in the commercial sector, and are planned for launch on geostationary communication satellites in 1999 and 2000 [1]. Figure 1 shows the Harris Corporation 12-m double-articulated radial-rib antenna. Figure 2 shows the TRW Astro 12-m perimeter-truss antenna.



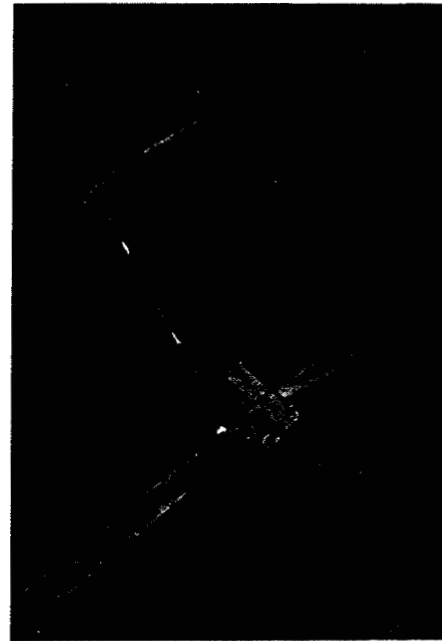
**Figure 1** 12-m double-articulated radial-rib antenna in test facility. (Courtesy of Harris Corp.)

The commercial-sector launches of the 12-m antennas will provide opportunities to observe the deployment reliability and performance in space of these new types of large lightweight antennas, with implications for their use in remote sensing science missions. The reflectivity and transmission loss characteristics of wire mesh antennas have been studied in detail [2]. However the use of these antennas for remote sensing requires additional analysis due to: (1) the unique requirements for radiometric precision and accuracy, and (2) effects of mechanical scanning on beam distortion and demands on spacecraft attitude control.

The reflective surface of mesh antennas consists of a wire mesh supported by a rigid rib or truss structure that can be folded compactly for launch and deployed in orbit. The mesh is made of gold-plated molybdenum wire typically of ~1.2-mil diameter knit into a diamond shaped pattern. A 10-opening-per-inch knit density is common which provides good reflectivity up to X-band. The nature of the knit permits the individual wires to shift relative to each other when not under tension. This allows the mesh to be stowed compactly and then stretched to the deployed configuration with minimal short wavelength surface error. The surface shaping mechanism is a tensioned cable or web network that provides a tie-down structure for the large number of flexible ties that interface to the mesh reflector. The larger the number of ties, the higher the surface precision. The mass of a typical 6-m reflector (excluding deployment boom and feeds) is less than 15 kg, and the RSS (root-sum-squared) error of the surface, accounting for all sources, is less than 0.6 mm. This gives a  $\lambda/20$  or better surface precision performance at frequencies up to 25 GHz.



(a)



(b)

**Figure 2** 12-m perimeter-truss antenna: (a) In test facility; (b) Artist's drawing, on geosynchronous communication satellite. (Courtesy of TRW Astro.)

In this study the following aspects are addressed:

*Requirements Analysis*—Definition and evaluation of instrument specifications and design, including error budgets and system trades.

*Antenna, Spacecraft, and Launch Vehicle Configuration Analysis*—Development and evaluation of preferred antenna, spacecraft, and launch vehicle configurations.

*Antenna and Feed Design*—Design of the reflector and feed configuration, and evaluation of the antenna beam performance.

*Mesh Radiometric Performance*—Laboratory measurements of mesh emissivity to predict radiometric performance of the reflector in the orbital thermal environment.

*Lightweight Radar Electronics*—Design of a lightweight radar system with mass, volume, and power estimates that meet the requirements of the space instrument.

In the following sections of this paper the design of the system is described and some results of the system analysis are presented. Much of the analysis work is still in progress.

## 2. SYSTEM DESCRIPTION

### *Science Requirements*

Science requirements for sea surface salinity (SSS) and soil moisture observations from space are outlined in reports of the Salinity Sea Ice Working Group (SSIWG) and the NASA Land Surface Hydrology Program (LSHP) [3], [4]. The SSS requirements are used as the basis for the system design described in this paper. A SSS measurement accuracy of 0.2 psu ('practical salinity units' or parts per thousand) at a 1-week, 100-km time-space scale is the primary target. These requirements can be met by a sensor with a footprint resolution of 40–60 km, an accuracy per footprint of ~0.6 psu, and a wide swath for providing global coverage in 2–3 days. Averaging the data on a 1-week, 100-km grid reduces the measurement noise error, and provides an accuracy of ~0.2 psu. Averaging on a 1-month, 200-km grid provides a potential accuracy of 0.05 psu. Depending on the correlation scales of the measurement errors, and by assimilating in situ data for calibration, further improvements in accuracy may be achievable. The technique for retrieving SSS from microwave measurements, including choice of radiometer and radar channels, polarizations, and incidence angle, and results of detailed simulations of the error sources and their impact on SSS retrievability, are discussed in [5].

The measurement precision and stability requirements for SSS are more demanding than for soil moisture, although the spatial resolution requirements for SSS are less demanding. The baseline system discussed here provides a spatial resolution of 40 km that meets the requirements for a "hydroclimate" soil moisture mission. Some soil moisture applications, however, require higher spatial resolution (~10 km) which will require a larger antenna than considered in the present study.

The system characteristics assumed for this study are derived from the science requirements and retrieval analysis discussed in [5]. A low cost (i.e. total mission cost less than about \$120M) 3-year science mission is the objective, as targeted towards an experimental mission in NASA's post-2002 Earth-probe mission scenario [6].

The SSS accuracy requirement of 0.2 psu leads to requirements for radiometric precision of 0.1 K and calibration stability of 0.2 K, radar precision and stability of 0.2 dB, incidence angle of  $>40^\circ$ , and a conical-scan (such that the incidence angle remains fixed across the swath). A beam-pointing knowledge of  $0.1^\circ$  is required to keep the corresponding brightness temperature uncertainty to less than 0.15 K. Pointing control to within about half a 3-dB-beamwidth is necessary for accurate geolocation. A polar, sun-synchronous orbit with pre-dawn equator crossing (5–6am) is required to obtain global coverage and to minimize Faraday rotation, and is advantageous from the

point of view of thermal stability of the instrument and utilization of solar power. The above requirements are addressed in more detail in [5]. In this paper we focus on the system design and analysis.

### *System Design*

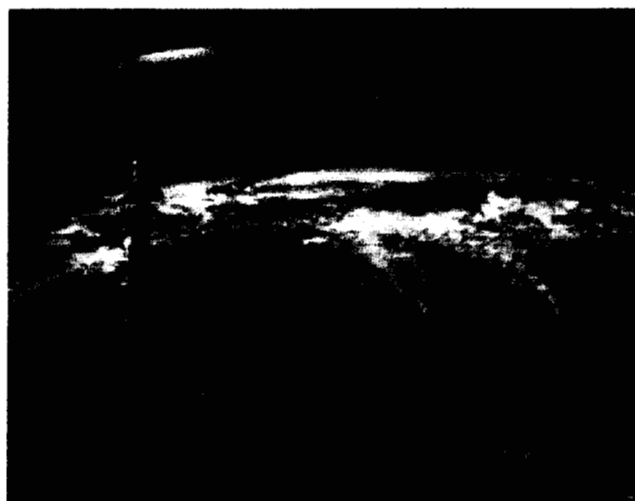
Implementation of a 6-m-aperture antenna in space, with multiple channels and the ability to scan over a wide swath, has earlier been considered a rather challenging and costly proposition. Recent developments, however, in mesh antenna technology have reduced the cost and risk, such that a system can be reasonably considered for an ocean salinity mission in the 2004 time frame. A spinning antenna system for precision radiometry is a new application for spaceborne mesh antennas, however.

Two configurations were considered for the OSIRIS application: (a) *Conical-Scan*—in which a wide swath is generated by rotating a parabolic reflector with a small number of feeds about a vertical axis; and (b) *Conical-Pushbroom*—in which the swath is generated by a large number of feeds, or a few scanning feeds, at the focus of a non-scanning parabolic-torus antenna. The conical-pushbroom concept has the advantage that the reflector itself does not scan. However the overall diameter of the reflector must be significantly larger, and the feed system and deployment more complex. The conically-scanning configuration is conceptually simpler and cheaper, but the rotating reflector places considerable demands on the spacecraft attitude control system and system reliability. On balance, considering technology readiness, cost, and risk, the conical-scan configuration was adopted for this study. A 6-m antenna was considered adequate to meet the science requirements.

Due to the limitations on mission cost, compact, low mass, and low power designs are required for the radiometer and radar electronics and feeds. Profiled corrugated horns are more massive but provide better antenna patterns than simple conical horns and allow more compact feedhorn design. Equal beamwidths at all channels are desirable so that all channels view approximately the same surface footprint. This improves the accuracy of the geophysical retrievals. The requirements on beam efficiency, antenna gain, and cross-polarization provide good sensitivity for the retrievals and avoid the need for elaborate antenna pattern corrections. Absolute calibration of the entire system in orbit is necessary, including the antenna reflector calibration and pointing control. This can be accomplished by occasionally rotating the entire system to provide a cold-space view, and by calibration against stable in-situ targets on the surface.

A number of configurations were considered for the orientation of the antenna and the position of the antenna relative to the spacecraft. Stability considerations dictate that the center of mass of the spinning system be on the vertical rotation axis. Due to the requirement to minimize the amount of solar radiation entering the feedhorns and reflector sidelobes during a full rotational scan of the antenna, a configuration with the feedhorns below the reflector was

chosen. A preliminary analysis indicated that for this configuration positioning the spacecraft below rather than above the antenna would result in a system that was simpler to deploy and control in orbit, and would place the radar and radiometer electronics close to the feeds and the spacecraft data system. Hence this configuration was adopted for the baseline system. An artist's depiction of this configuration is shown in Figure 3.



**Figure 3** Artist's concept of OSIRIS

#### *System Characteristics*

The baseline system adopted for our study uses a 6-m-diameter conically scanning antenna. The key system characteristics are summarized in Table 1. The antenna system is a rotating, offset-fed, parabolic-mesh reflector, with two identical multichannel feedhorns. The two feedhorns provide separate beams that provide increased sampling and allow the antenna system to rotate half as fast as would be necessary with a single beam. The combined antenna and feed system rotates about the vertical axis, with antenna beams offset at  $\sim 36^\circ$  from nadir, providing a wide-swath conical scan. As the spacecraft moves, the 3-dB antenna footprints provide overlap along and across track in a helical coverage pattern. The rotation rate of 6 rpm with two beams provides overlapping contiguous footprints at the surface. At an orbit altitude of 600 km, the 6-m antenna provides about 40-km spatial resolution, an incidence angle of about  $40^\circ$ , and a swath width of 900 km. (At an altitude of 800 km, the corresponding parameters are 56-km spatial resolution,  $42^\circ$  incidence angle, and 1200 km swath width.) A low orbit is preferred from the point of view of spatial resolution. However, at orbit altitudes lower than about 600 km atmospheric drag becomes a concern, requiring more attitude control and orbit maintenance, and increased fuel.

Mission cost constraints require that the stowed volume of the spacecraft and payload fit within a Taurus-class or smaller launch vehicle. The spinning antenna requires a spacecraft that either rotates with the antenna as a rigid body or is 3-axis stabilized with a spinning platform on which the antenna is mounted. The large spinning antenna places

demands on the spacecraft particularly in the area of attitude control in order to meet the requirements for pointing control and knowledge. The set of baseline characteristics are shown in Table 2.

**Table 1.** Key baseline system characteristics

Radiometer frequencies	1.41 and 2.69 GHz
Radiometer polarizations	H, V; (1.41 GHz polarimetric)
Radar frequency	1.26 GHz
Radar polarizations	VV, HH, VH, HV
Antenna type	Offset-fed, parabolic, deployable mesh reflector
Aperture diameter	6 m
Nadir offset angle	$36^\circ$
Number of feedhorns	2 (each L/S-band, V/H-pol)
Beamwidths	$2.6^\circ$ (approx. equal all channels)
Antenna gain	35 dB
Beam efficiency	$> 90\%$
Cross-polarization	$< -18$ dB
Orbit type	Polar, sun-synchronous, 6am/6pm
Altitude	600 km
Spatial resolution	35 x 45 km
Swath width	900 km
Rotation rate	6 rpm
Global coverage	2-3 days
Pointing control/knowledge	$1.3^\circ/0.1^\circ$ ( $3\sigma$ )
Radiometer precision/stability	0.2 K
Radar precision/stability	0.2 dB
Data rate	25 Kbits/sec
Launch vehicle	Taurus-class
Mission duration	3 years
Mission cost	$\sim \$120$ M

#### *System Options*

Two options were considered as alternatives to the baseline system for reducing the potential cost of a mission for which high spatial resolution and wide-swath scanning may be of lesser importance.

1. A 4-m-diameter scanning antenna that would maintain wide swath coverage but reduce the spatial resolution from 40 to 60 km.
2. A 4-m-diameter non-scanning antenna that would reduce the spatial resolution from 40 to 60 km and would sample along a single footprint, taking a month to achieve global coverage at a 100-km grid scale.

These options are considered as fall-back designs if the cost of the baseline system proves prohibitive and the reduction in science yield is acceptable.

**Table 2.** IIP baseline mission mass and power summary

	Mass (kg)	Feed mass (kg)	Power (W)
Mesh antenna & boom	45		
Feedhorns (2)		4	
Radiometers		3	24
Backend electronics	10		20
Radar front end (spin)		3	90
Radar C&DH (spin)	15		25
Radar C&DH (back end)	20		25
Spin assembly*	50		25
Total	140	10	209

\* For 3-axis stabilized spacecraft option

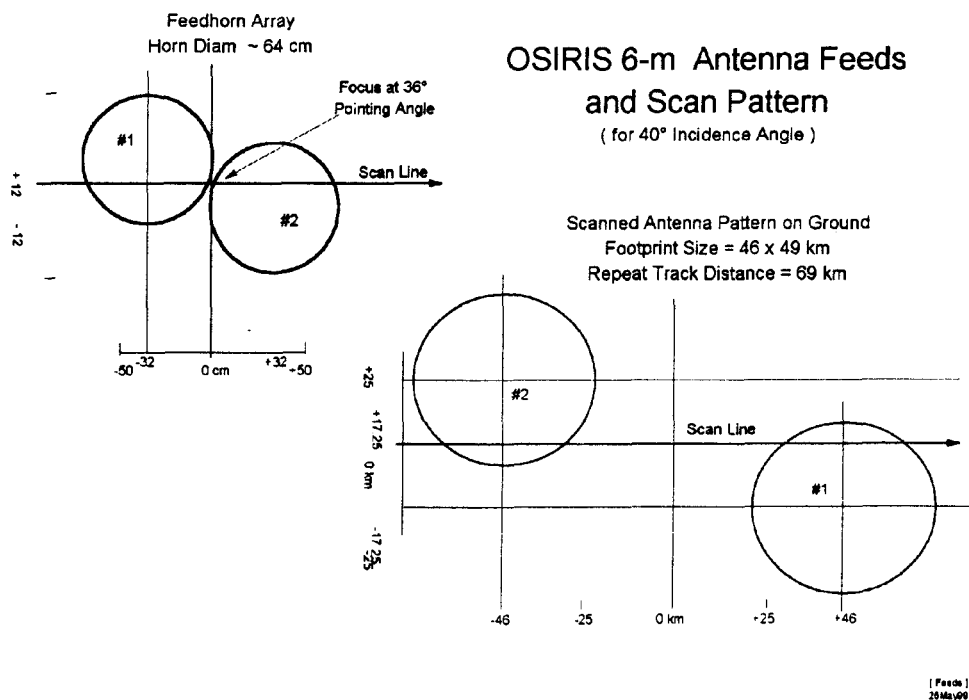
### Sampling and Data Rate

For a mapping system it is usually required that the antenna footprints of adjacent samples overlap at least at the 3-dB level both along and across track. This requirement determines the sensor sampling rate and the antenna scan rate for a given orbit altitude, antenna diameter, and number of beams. A preliminary analysis indicated that no more than two feedhorns could be accommodated on a small

spacecraft since each feedhorn at L-band has a diameter of about 0.64 m. Figure 4 shows the design for the positions of the feedhorns, placed adjacent to each other with centers offset by 12 cm to either side of the scan line. This feed placement provides the antenna footprint spacing shown, with centers offset by 17.25 km to either side of the scan line and a designed 25% overlap of the 3-dB footprints in the along-track direction. The footprints are shown as approximately circular since the radiometer integration time smears out the short dimension of the elliptical footprint in the along-scan direction to make the effective footprint shape of each sample approximately circular.

OSIRIS is a coarse-resolution system with a relatively low data rate. It is advantageous for the radiometer and radar data system to sample at a higher rate than required for 3-dB footprint overlap in the along-scan direction. Subsequent ground data processing can be used to average the samples to obtain approximately co-registered footprints at all channels. Given these sampling design characteristics, the required antenna spin rate, sensor integration times and data rates were computed. The spin rate was determined as 6 rpm. Parameters relevant to the data rate calculation are given in Table 3 which shows a total data rate for the system of 25.6 Kbps.

Various scenarios were investigated to determine the data downlink and on-board storage requirements for the baseline system. The scenarios considered the OSIRIS orbit, data rate, and desire to consider a data latency of as low as six hours, requiring as many as four downlinks per day.



**Figure 4** OSIRIS feedhorn placement and footprint spacing.

**Table 3. Sampling and data rate**

Antenna spin rate	6 rpm
Data sampling rate	100 Hz
Sample spacing along scan	2.83 km
Integration time per footprint	124 ms
Word length (includes overhead)	2 bytes
Number of radiometer channels	6
Number of radar channels	4
Number of beams	2
Radiometer data rate	19.2 Kbps
Radar data rate	6.4 Kbps
Total data rate	25.6 Kbps (2.2 Gbit/day)

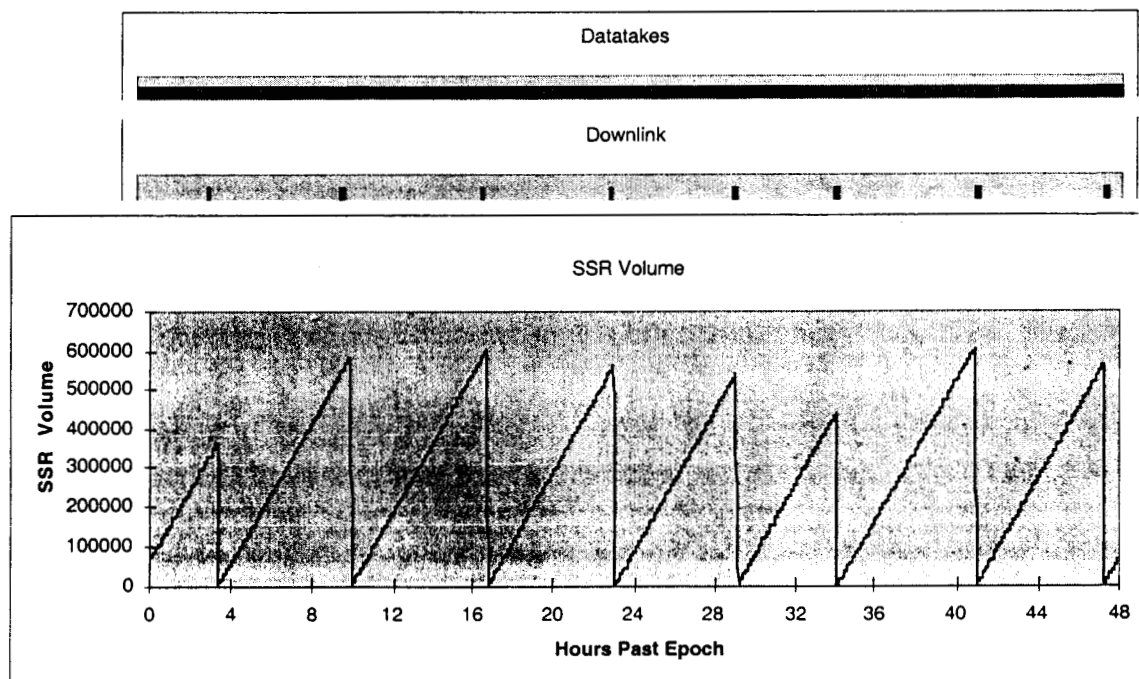
Figure 5 shows the downlink timing, and build-up and playback of data on the spacecraft solid-state recorder (SSR), for a 4-pass-per-day scenario. The top bar illustrates continuous data acquisition by the radiometer and radar sensors. Each tick mark on the downlink bar indicates an available overpass of either the NASA Wallops or Alaska ground station. There are more passes available than shown. Only selected passes, spaced approximately six hours apart, are shown. The vertical axis on the lower graph shows the volume in Kbits. For this scenario, an on-board SSR storage of 0.6 Gbit and a 1.2 Mbps S-band link to a 5-m ground station antenna is sufficient. The ground station cost decreases for a 2-pass-per-day scenario since the number of

passes tracked is lower. The cost increases again for a 1-pass-per-day scenario due to the need for an X-band downlink to accommodate the higher transmission rates necessary to downlink a full day of stored data in one pass.

#### Calibration Requirements

A set of calibration requirements has been developed to guide the analysis of the system. Table 4 shows these requirements in the form of a target system error budget. It is assumed that absolute calibration biases can be corrected by comparisons of the radiometer and radar observations with in-situ data, and through adjustment of the geophysical model functions. The measurement performance is then limited by the temporal stability of the calibration. The rationale for the target error budget is given below.

Errors due to non-ideal antenna patterns can be limited to less than 0.1 K for beam efficiencies of greater than 90%. The antenna reflector mesh shaping and support structures are constructed of composite and temperature-insensitive materials that are thermally stable in orbit. In addition, the tension of the mesh is designed to minimize shape-distortion due to thermal expansion. An antenna beam-pointing error of 0.1° should be feasible given careful attention to spacecraft attitude control. An internal reference load and highly stable noise diode source are used for the radiometer calibration. An internal calibration loop is used for the radar. The stability of the noise source for radiometer calibration has been demonstrated for the JASON (TOPEX follow-on) microwave radiometers. The stability of the radar calibration loop has been demonstrated by testing of the



**Figure 5** Data downlink and storage volumes for 4-pass-per-day scenario (volume in Kbits).

**Table 4.** Calibration requirements summary

Parameter	Stability (1 sigma)			Bias (3 sigma)		
	Parameter Error	Radiometer (K)	Radar (dB)	Parameter Error	Radiometer (K)	Radar (dB)
Antenna beam gain & pattern		0.1	0.15		2	0.7
Antenna beam pointing	0.1°	0.15	0.1	0.3°	0.45	0.3
S/C attitude	0.01°	0.02	0.01	0.03°	0.05	0.03
Calibration noise source	0.1 K	0.1	0	1 K	1	0
Radar calibration loop	0.05 dB	0	0.05	1	0	1
Waveguide/coax cable loss	0.05 dB	0.05	0.01	0.1 dB	0.1	0.1
RSS		0.2	0.19		2.5	0.9

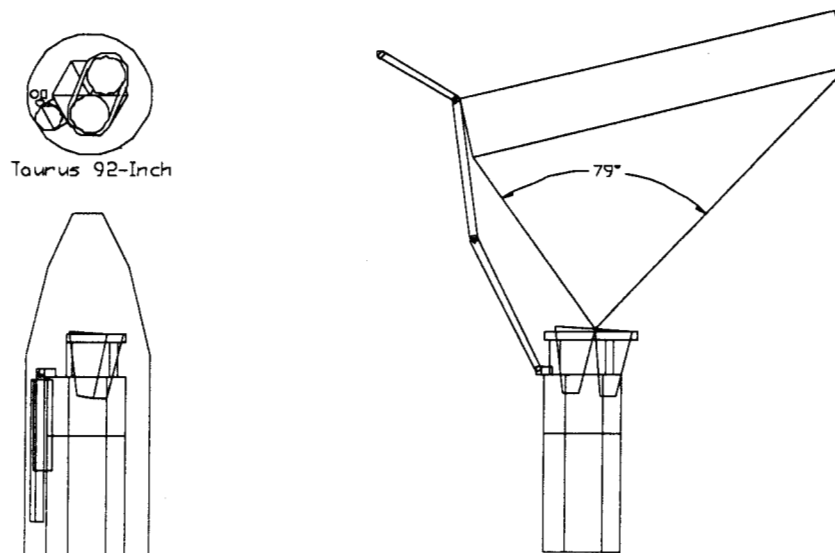
SeaWinds/Quikscat scatterometer launched in June 1999. The effects of the losses in the antenna reflector and feed horns can be corrected by careful temperature measurements of these components. In summary, the RSS calibration stability requirements of 0.2 K and 0.19 dB for the radiometers and radar, respectively, are challenging but feasible.

### 3. CONFIGURATION ANALYSIS

Candidate antenna, spacecraft and launch vehicle configurations have been studied to evaluate the stowed volume, mass, deployment and attitude control requirements of the system. Figure 6 shows one possible configuration of the stowed and deployed antenna system. The configuration shown is for a 6-m TRW Astro antenna, a TRW SSTI Core spacecraft, and a Taurus launch vehicle with 92-inch fairing. This configuration is illustrative but does not imply an optimized or selected system. The 92"-Taurus is capable of placing about 720 kg of payload into a 600-km sun-synchronous orbit. The configuration shown,

with the antenna mounted above the spacecraft, was selected as the best design based on simplicity of deployment, minimum attitude control system (ACS) requirements, design flexibility, and environmental impacts (e.g. solar heating, solar leakage to feedhorns, environmental momentum).

Two alternatives for the spacecraft were considered, one a rigid-body spinner with the antenna directly mounted and the other a non-spinning 3-axis stabilized bus with a spinning platform for the antenna. Preliminary torque and momentum analyses indicate that the system will operate best at zero momentum with a large momentum wheel to counteract the 6-rpm rotation of the spacecraft and antenna (in the rigid-spinner configuration) or antenna only (in the 3-axis-stabilized case). The momentum storage requirement is likely to be in the range from 120 N-m-s (3-axis) to 200 N-m-s (rigid-spinner). A large momentum storage requirement affects the transverse momentum storage requirement and tolerances on spin balancing.



**Figure 6** Illustration of antenna and spacecraft in stowed and deployed configuration

Analysis of environmental forces (i.e., solar pressure and aerodynamic torque) shows that a large amount of momentum will be dumped into the momentum control system. This is due to the large separation between the center of pressure and the center of gravity. Using magnetic torquers to continuously dump momentum will still leave about one fourth of the orbit momentum. For the rigid spinner configuration, this implies that reaction wheels must have a large torque authority (0.3–0.6 N-m) to precess accumulated momentum at 6 rpm. For the 3-axis-stabilized spacecraft, the precession rate is the orbit rate, i.e. about 1/90 rpm. Hence, the required torque authority is smaller.

A rigid-spinner spacecraft will have stringent ACS requirements. There is also less power available due to the limited surface area for solar cells (although this should not be a problem for OSIRIS). A 3-axis-stabilized spacecraft requires a spinning platform for the payload and deployment mechanisms for the solar arrays. Both add to the mass of the spacecraft as well as reduce reliability. A rigid-spinner spacecraft is thus expected to be lower mass, more reliable, and less complex and expensive. The choice between the two spacecraft options is not clear-cut, and further analysis of the ACS system is needed to determine an optimized spacecraft configuration.

#### 4. ANTENNA AND ELECTRONICS DESIGN

Analyses have been done of antenna and electronics subsystems designs for the baseline OSIRIS system. For the antenna system, conical, conical-corrugated, and profiled-corrugated horns were evaluated for their performance in terms of far-field pattern symmetry, beam efficiency, and cross-polarization. Effects of the displacement of the feeds at the reflector focus and reflector surface distortions were also analyzed.

The profiled or compact corrugated horn was found to have the most suitable symmetrical-beam performance over the

large frequency range required (1.2–2.7 GHz). Figure 7 shows the far-field patterns using this horn design, with the feeds placed off-focus as shown in Figure 4. The beam efficiencies are 94.3% and 92.1% at L and S band, respectively. The S-band beam is more asymmetric than the L-band beam since the feed design was optimized for L band (given the greater importance of the L-band measurement channels).

The effects of random surface error were evaluated in terms of the rms error tolerable for a beam efficiency of no less than 90%. At L band the error is  $\lambda/55$  or 3.9 mm, while at S band the error is  $\lambda/80$  or 1.4 mm. This is well within the expected capability of current mesh reflectors as discussed in Section 1. However, further work is needed to characterize the effects of anticipated thermal and dynamical distortions of the antenna system in orbit.

For a spinning antenna system it is critical to keep the system mass low to ease the requirements for spacecraft momentum compensation. There is greater potential for mass savings in the radar than radiometer design, hence attention was focused on the radar electronics. A radar design with advanced space-qualified electronics was developed to obtain mass, volume and power estimates.

Key aspects of the radar design include a 100-W RF transmit power and 4-MHz chirp bandwidth to obtain better than 0.2 dB Kpc. The radar pulse length is 1 ms (10% duty cycle) which enables 90% of the time to be available for the radiometer integration. A fixed range gate of 1.24 ms can accommodate the variation of the echo arrival time for an elliptical earth. A polarimetric capability is included to enable correction of Faraday rotation. A detailed electronics layout and parts selection was done to ensure reliable estimates of mass and power. The total size of the radar electronics package is estimated as 22 x 22 x 9 cm. The mass is estimated as 26 kg and the power required is 127 W.

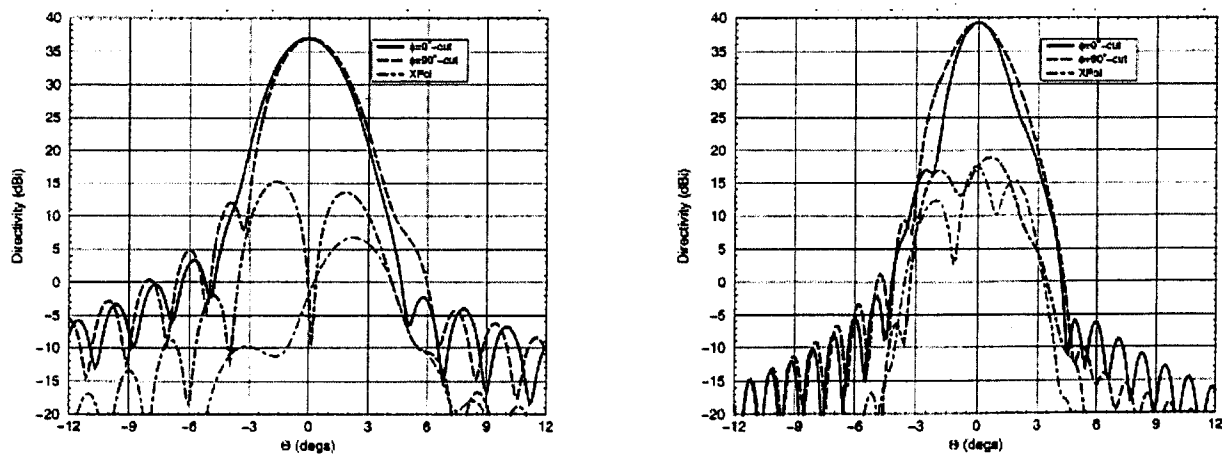


Figure 7 Far-field antenna patterns at (a) L band, (b) S band.

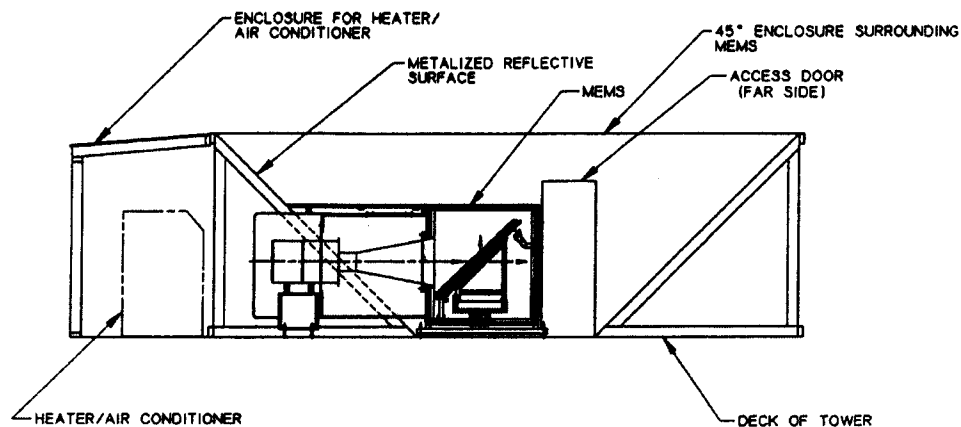


Figure 8 Mesh Emissivity Measuring System

## 5. MESH MEASUREMENTS

Although deployable mesh antenna technology has been used successfully for many years for spaceborne telecommunications applications, its use for microwave radiometry applications is not as well developed. For radiometry the emissivity of the reflector is extremely important, and losses that would be of little consequence in a communications application can have dire effects for remote sensing. The variability of this loss or emissivity of the reflector, and not its absolute value, is the major concern, and must be carefully considered to take optimum advantage of this technology for remote sensing.

The baseline OSIRIS error budget allocates no more than 0.1 K for variations in the radiometric properties of the mesh reflector. A study has been initiated to measure directly the radiometric properties of relevant mesh samples and to ensure that the radiometric performance of the mesh is consistent with the mission objectives.

Figure 8 shows the Mesh Emissivity Measurement System (MEMS) under development at the NASA Langley Research Center. The measurement setup is an advancement of the "Sky Bucket" approach used for mesh measurements in the early 1980's for a previous radiometric mesh antenna system proposed by the Naval Research Laboratory. The mesh emissivity measurement goals for this system are 0.001 and 0.0002 for accuracy and precision, respectively. Characterization of the system is currently underway and results will be available during the coming year.

## 6. CONCLUSIONS

This paper has provided an overview of the status of a study of a large mesh antenna system for spaceborne microwave sensing at L and S bands. A baseline system has been defined and evaluated from the standpoints of science accuracy requirements, preferred antenna, spacecraft and launch vehicle configurations, and antenna and electronics designs. Additional studies are being performed to

characterize the radiometric properties of the mesh, refine the system design, and to provide detailed simulations of the expected system performance in orbit.

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